BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Potential Use of the CSM for Lunar Orbital Mission Work
Case 232

DATE: June 4, 1967

FROM: C. J. Byrne
W. L. Piotrowski
D. R. Valley

ABSTRACT 28047

This memorandum deals with the potential application of the CSM for orbital mission work after it has assisted in landing an unmanned Surface Payload Module. Limits to the CSM inert weight imposed by the launch vehicle capability and Service Module propulsion system have been determined. The effect of trading CSM inert weight growth for Service Module propellant for orbital maneuvering is investigated as is the effect of jettisoning equipment in lunar orbit.

Some of the possible spacecraft configuration and mission modes are discussed briefly. If the primary flight objective of landing the unmanned Payload Module is not to be compromised, a low inclination orbital mission appears the most feasible.

Remote sensing experiments which are applicable to such a low inclination orbit are discussed and a list of instruments required for these experiments is suggested. It is pointed out that more than 20 near overflights of the center of Copernicus can be obtained with a 10° inclination and the proper selection of the point of nodes, making a low inclination orbit particularly attractive for remote sensing of certain areas of scientific interest.

Author

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MEMORANDUM FOR FILE

INTRODUCTION

Many lunar exploration concepts propose unmanned logistics flights using a LM derivative landing vehicle. The operational mode for these flights uses a manned CSM to assist in the unmanned landing of the cargo vehicle. After the landing is accomplished, the manned CSM can immediately return to earth or can remain in lunar orbit to conduct orbital experiments and reconnaissance before returning to earth. The latter alternative appears attractive; however, additional equipment for the orbital work would have to be carried.

WEIGHT

The resulting weight increases to the CSM must be held within the capabilities of the launch vehicle and the Service Module propulsion system. Appendix A contains a general solution to the problem of determining the allowable CSM weight increases for lunar orbital applications. The solution also includes the possibility of trading CSM inert weight against SM propellant for orbital maneuvering as well as the effect of jettisoning some of the added equipment in lunar orbit prior to trans-earth injection.

The results of calculations using the general solution of Appendix A are plotted in Figure 1 for the following two specific cases:

CASE I - Fully loaded Block II SM propellant tanks (38,986 lbs) and a Saturn V injection capability up to 102,000 lbs.

CASE II - Saturn V injection capability fixed at 98,000 lbs with variable SM propellant loads as limited by the injection capability.

[redacted]

The following basic assumptions were used in calculating the results plotted in Figure 1:

1. LM separation weight of 32,500 lbs. (Thus no sacrifice to the unmanned lander's payload.)

2. Apollo LOR type mission ΔV budget excluding the allotment for LM rescue.

3. Added inert weight is considered to be jettisoned prior to entry into earth's atmosphere. (Special cases are also examined for partial jettison in lunar orbit.)

The allowable CSM inert weight in Case I, for example, could be as high as 25,938 lbs. Based on proposed new CSM control weights (23,562*), 2,376 lbs could be added to the CSM. Weight additions of this magnitude bring up the question of a trade with Service Module propellant for orbital maneuvering. Figure 1 indicates this type of trade in a plot of CSM inert weight against Service Module propellant weight for maneuvering. In Case I, reducing the CSM inert weight from 25,938 lbs to 24,452 would leave 1000 lbs of Service Module propellant for orbital maneuvers. This amount of propellant would provide a ΔV capability of almost 300 ft/sec which could produce about 3° plane change.

The dashed line associated with the Case I plot shows the effect of jettisoning 1,000 lbs of equipment while in lunar orbit rather than waiting until just prior to earth entry. It can be seen that lunar orbital jettison of 1,000 lbs would allow increasing the CSM inert weight (prior to equipment jettison) by 550 lbs, or else would make 372 lbs additional Service Module propellant available for orbital maneuvering.

Figure 1 also shows a similar plot for Case II - a 98,000 lb Saturn V with variable Service Module propellant loadings as limited by the launch vehicle injection capability. Elimination of the LM rescue requirement allows the CSM to grow to 24,300 lbs. The trade-off with orbital maneuvering propellant and the benefits of lunar orbit jettison can be seen to differ from Case I. The difference is due to the Service Module propellant loading limitation imposed by the 98,000 lb Saturn V. The general solution (Appendix A) will help to explain the different slope in the plots for the two cases as well as the different CSM weight tradeoff indicated for lunar orbit jettison (1,000 lbs jettisoned allows CSM inert weight increase of 271 lbs for Case II as compared to 550 lbs for Case I).

*Includes 3 man crew.

CONFIGURATION

Three possible configurations for orbital instruments are shown in Figure 2. Configuration A constrains all instruments to be carried within the Command Module. Data would be obtained through the CM windows or by deployment of the instrument through the docking hatch. Some instruments could be stored in the small volume ($\sim$ 3 ft^3) reserved for the sample return boxes, and if the crew were reduced from 3 men to 2 for a logistics mission, the volume of the third crewman's station could also be used. This would provide ample volume plus an additional weight allowance on the order of 1,000 lbs.

Configuration B places the instruments in Sector I of the Service Module. The volume available in this location is dependent on the mission duration since the additional expendables required would probably be placed in this sector. Aside from this trade-off, the following considerations make this an undesirable location for the orbital instrumentation:

1. Experience with the LM&SS and Pallet indicates a considerable interface task.

2. Modifications to the Service Module Structure (windows) would require an expensive program for requalification.

3. Access for equipment deployment and data retrieval would be difficult.

4. Lunar orbital jettison of the equipment would be difficult.

Configuration C uses an intermodule between the CM and the unmanned Surface Payload Module (stripped ascent stage version). The intermodule is attached to the docking ring of the ascent stage during launch and is transferred to the CSM after transposition and docking. Explosive bolts separate the surface payload module while the intermodule remains attached to the CSM docking probe for use in lunar orbit. The intermodule* can provide 120 ft^3 for instrumentation which could accommodate up to 3000 lbs.

If the LM truck version of the Surface Payload Module is considered it would be possible to use the end of the Payload

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*Systems Description of Modular Apollo Extensions (M.A.E.) Lockheed Missiles & Space Company - March 20, 1967.

Module (used with the Lunar Mapping and Survey System) as an intermodule. This would permit carrying over the PM remote sensors with minor modifications. In this configuration, the LM truck payload envelope would have to be shortened, or built with a recess in the top to fit the end of the PM. The PM docking apparatus would be used, leaving the truck free of this requirement. A new structure would be needed to carry the LM truck loads around the intermodule to the docking collar.

In each configuration, it may be desirable to include a subsatellite which would provide its own power, altitude control and communication. The subsatellite could be specified to be free of radiation, magnetic fields, and vibrations.

MISSION

Since the primary mission would be landing of the Surface Payload Module, it is realistic to assume a highly constrained orbital mission. Considerations of the surface rendezvous mission would constrain the landing site to the equatorial band on the near face of the moon.* Although it would be possible to go into polar orbit and wait about 10 days to land the Surface Payload Module in the proper location, this delay would compromise success of the primary mission. Therefore, this mission would probably be at low inclination (of the order of 10° or less).

Such a low inclination mission (<10°) does not lend itself to broad systematic surveys but allows for near overflight of a broad swath in the equatorial region. Figure 3 shows the first, twelfth, and twenty-fourth ground tracks for a 10° inclination with the twelfth track centered on Copernicus. Consecutive ground tracks are separated by a distance of 5.2 km at the equator and the separation decreases as latitude increases, finally crossing near 10°. A sensor beam width greater than 3.7° (from an orbital height of 80 km) allows for overlap at this maximum separation. The ground tracks show that for a node at 70° E and a 10° inclination, the center of Copernicus will be visible from the CSM for considerably more than 20 orbits. Assuming that the CSM stays in lunar orbit 6 days, approximately 42% of the region between 10° N and 10° S can be surveyed from orbit with sensor beam width of 3.7° or greater. The areas near 10° latitude will be repeatedly near overflown, permitting the sensors to view approximately the same areas and improve the counting statistics for certain experiments.

EXPERIMENTS AND PAYLOAD

At this phase in the Lunar Exploration Program (landing an unmanned SPM) several manned landings will have taken place near the equator and surface scientific stations

*This assumes that an abort-anytime constraint is to be maintained.

will be collecting data on the geophysical and environmental parameters of the Moon. Ground truth data will be available for each landing site in addition to real time information on the surface magnetic field and incident electron and proton fluxes.

From the CSM in orbit it will be possible to:

1. correlate magnetic field measurements made at orbital altitudes with corresponding measurements made on the surface;

2. determine the lunar environment at orbital altitudes;

3. photograph the surface at various lunar phases;

4. measure the lunar surface radioactivity while the incident radiation varies;

5. determine the geometric shape and gravitational anomalies of the Moon;

6. determine the surface temperature and surface thermal properties;

7. establish an upper bound to the lunar heat flow;

8. determine the electrical conductivity and complex dielectric constant of lunar material;

9. construct a three dimensional brightness temperature map; and

10. determine the feasibility of multispectral photography for remote sensing.

To perform the measurements suggested above, the following experiments are recommended:

Cameras

Magnetometer

IR Radiometer (or Imager)

Multichannel Microwave Radiometer (or Imager)

Gamma-Ray Spectrometer

X-Ray Fluorescence

Radar Altimeter

Gravity Gradiometer

Multispectral Photography Feasibility Study

Electron Proton Spectrometer

Nuclear Emulsion Stacks

Micrometeoroid Collection Plates

There may be problems associated with the inclusion of the magnetometer due to the magnetic environment of the spacecraft and problems with isolating the gravity gradiometer from the effects of crew movement; the spacecraft radiation background may interfere with the gamma-ray spectrometer and X-ray fluorescence.

The results expected from the above instruments are discussed elsewhere.* The micrometeoroid collection plates and nuclear emulsion stacks are suggested for the CSM primarily because of the requirements of long exposure at large distances from the earth and return of the plates and emulsions to earth for analysis; the data from the plates and emulsions are expected to be relatively independent of the orbital inclination.

The mission profile for experimental purposes should consist of the following:

1. After landing the SPM, the CSM remains in an equatorial orbit for about 5 orbits to fly over the ground truth sites and calibrate the sensors;** and,

2. A plane change is effected with the point of nodes selected so that the CSM overflies the prime scientific sites a maximum number of times.

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*W. L. Piotrowski and B. E. Sabels, "Lunar Orbital Experiments with the Apollo Mapping and Survey System," Bellcomm, Inc., Memorandum for File, January 27, 1967.

**Repeated overflight of the same sites gives a direct determination of the repeatability of the measurements.

CONCLUSIONS

If a manned mission is used to deliver a surface payload module to lunar orbit, the CSM could perform useful orbital work. A choice of configuration should await further definition of the Surface Payload Module.

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Attachments:
Appendix A
Figures 1-3

APPENDIX A

Determination of allowable CSM inert weight growth for lunar orbital applications with unmanned delivery.

CSM = Inert weight of Command/Service Module

X = Weight jettisoned in Lunar Orbit

SMP_{LO} = Service Module propellant available for orbital maneuvers

R_1 = Mass ratio associated with trans-earth injection ΔV (3,190 fps)

R_2 = Mass ratio associated with lunar orbit insertion ΔV (3,607 fps)

SMP_T = Total Service Module propellant load

LM = Separation weight of unmanned lander (32,500 lbs)

SLA = S/C LV adapter (3,850 lbs)

<u>General Solution for Allowable CSM Weight Growth</u>:

$$CSM + SMP_T + LM + SLA = \text{Injection weight}$$

$$CSM + SMP_T + 32{,}500 + 3{,}850 = \text{Inj. wt. (Launch Vehicle Capability)} \tag{1}$$

$$\frac{CSM + SMP_T + 32{,}500}{R_2} = \text{Spacecraft weight in Lunar Orbit}$$

$$SMP_{LOI} = (CSM + SMP_T + 32{,}500)\left(\frac{R_2 - 1}{R_2}\right) = \text{SM Propellant used for Lunar Orbit Insertion}$$

$$SMP_{TEI} = (CSM - X)(R_1 - 1) = \text{SM Propellant used for Trans-earth Injection}$$

$$SMP_T = SMP_{LOI} + SMP_{TEI} + SMP_{LO}$$

NOTE: Apollo ΔV Budget taken from "Apollo Lunar Landing Mission Symposium - MSC - June 25-27, 1966"

$$SMP_T = [CSM + SMP_T + 32{,}500]\left[\frac{R_2 - 1}{R_2}\right] + [CSM-X]\ [R_1-1] + SMP_{10}$$

Solving for SMP_T

$$SMP_T = [CSM]\ [R_1R_2-1] - X[(R_1-1)(R_2)] + SMP_{LO}(R_2) + 32{,}500\ (R_2-1)$$

Using Apollo values for R_1 and R_2

(2) $$SMP_T = [CSM]\ [.963] - X[.532] + SMP_{LO}(1.431) + 14{,}007$$

Equations (1) and (2) represent a general solution to the problem of determining the CSM inert weight allowable for orbital application. These equations will now be applied to two specific cases to illustrate their application.

Case I - BLK II SM propellant tanks fully loaded (38,986 lbs) and Saturn V injection capability up to 102,000 lbs.

Equation (2) is all that is required since the total SM propellant (SMP_T) is always 38,986 lbs;*

$$38{,}986 = [CSM]\ [.963] - X[.532] + SMP_{LO}\ (1.431) + 14{,}007$$

If X = 0 (No weight jettisoned in lunar orbit), and

SMP_{LO} = 0 (No SM propellant for orbital maneuvers)

CSM = 25,938 lbs (See Fig. 1)

If X = 0 and SMP_{LO} = 1,000 lbs, CSM = 24,452

" " " = 2,000 lbs, CSM = 22,965

(The above results are plotted on Fig. 1 (Solid line for Case 1))
If X = 1,000 lbs (weight jettisoned in lunar orbit) and

SMP_{LO} = 0 ; CSM = 26,490 lbs

SMP_{LO} = 1,000; CSM = 25,004 lbs

SMP_{LO} = 2,000; CSM = 23,518 lbs

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*The Saturn V injection capability required is 75,336 lbs plus the CSM inert weight. (SMP_T + LPM + SLA or 38,986 + 32,500 + 3,850)

These results plotted are the dashed line for Case 1 on Fig. 1.

Case II - 98,000 lb Saturn V and variable SM propellant load as limited by launch vehicle capability.

Equations (1) and (2) are required for this case:

(1) $CSM + SMP_T + 32{,}500 + 3{,}850 = 98{,}000$ (Injection Wt)

(2) $SMP_T = [CSM]\,[.963] - X[.532] + SMP_{LO}(1.431) + 14{,}007$

Substitution (1) into (2)

$$61{,}650 - CSM = [CSM]\,[.963] - X[.532] + SMP_{LO}\,(1.431) + 14{,}007$$

$$47{,}643 = [CSM]\,[1.963] - X[.532] + SMP_{LO}\,(1.431)$$

If $X = 0$, and: $SMP_{LO} = 0$; CSM = 24,270 lbs

" " $SMP_{LO} = 1{,}000$; CSM = 23,541 lbs

" " $SMP_{LO} = 2{,}000$; CSM = 22,812 lbs

(These results plotted on Figure 1 (Case II Solid line))

If $X = 1{,}000$ and $SMP_{LO} = 0$; CSM = 24,542

" " $SMP_{LO} = 1{,}000$; CSM = 23,813

" " $SMP_{LO} = 2{,}000$; CSM = 23,084

(These results are plotted on Figure 1 (Case II Dashed line))

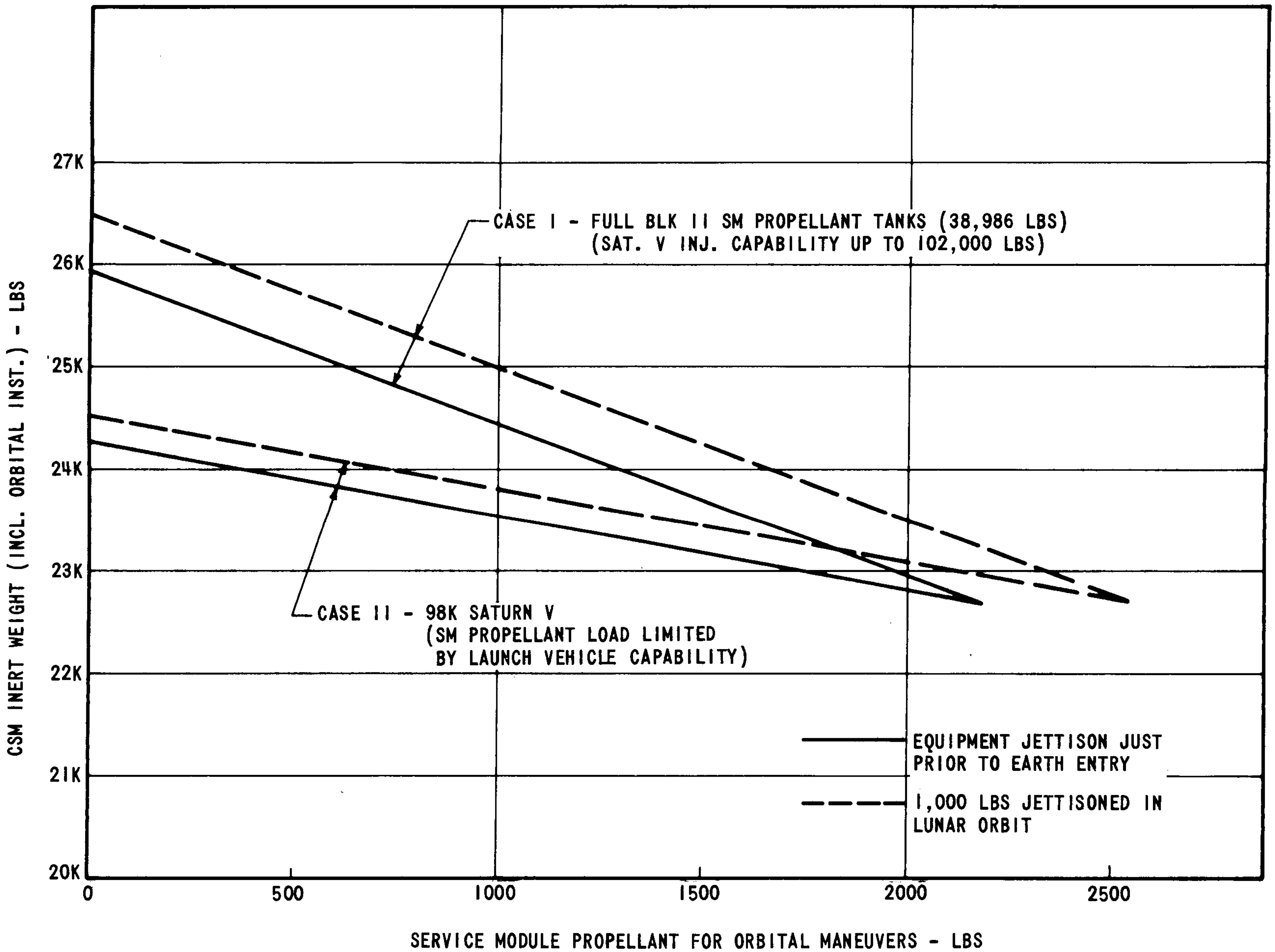

FIGURE I - ALLOWABLE CSM WEIGHT FOR LUNAR ORBITAL APPLICATIONS
(APOLLO ΔV BUDGET EXCLUDING LM RESCUE & 32,500 LPM SEPARATION WT.)

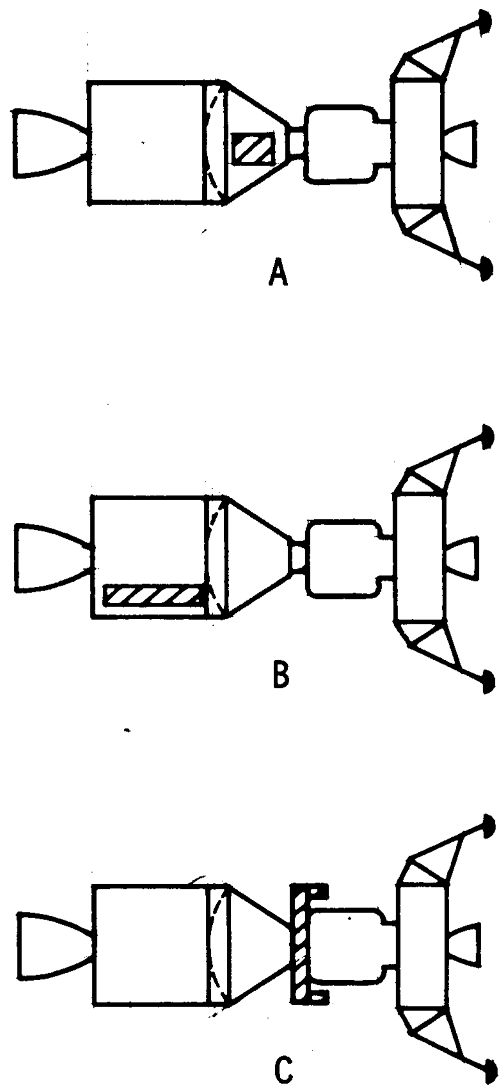

FIGURE 2 - CONFIGURATIONS

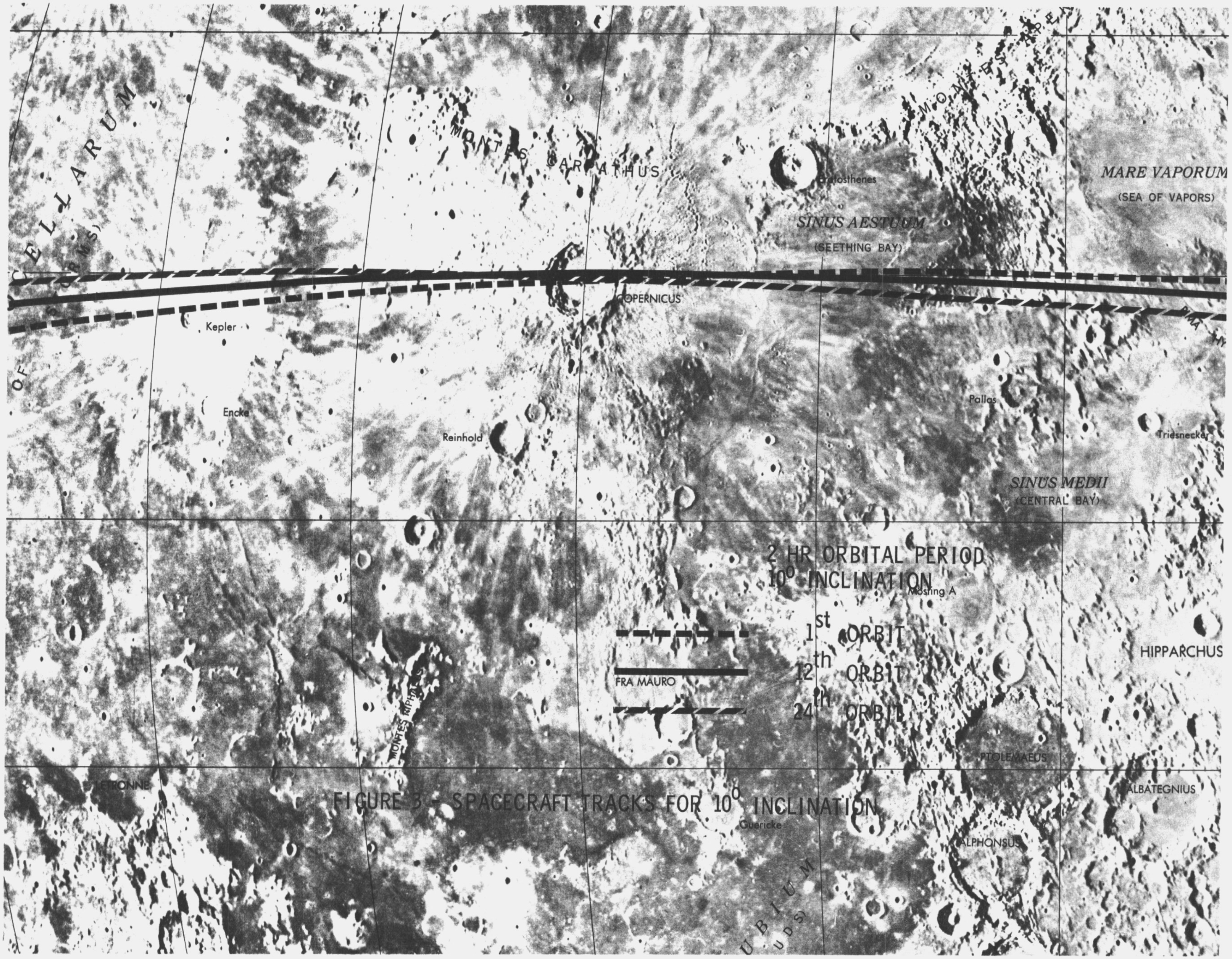

FIGURE 3 - SPACECRAFT TRACKS FOR 10^{0} INCLINATION

BELLCOMM, INC.

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